NASA-TM-108604

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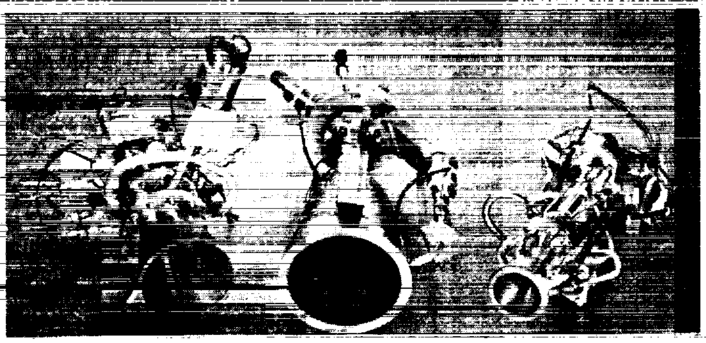
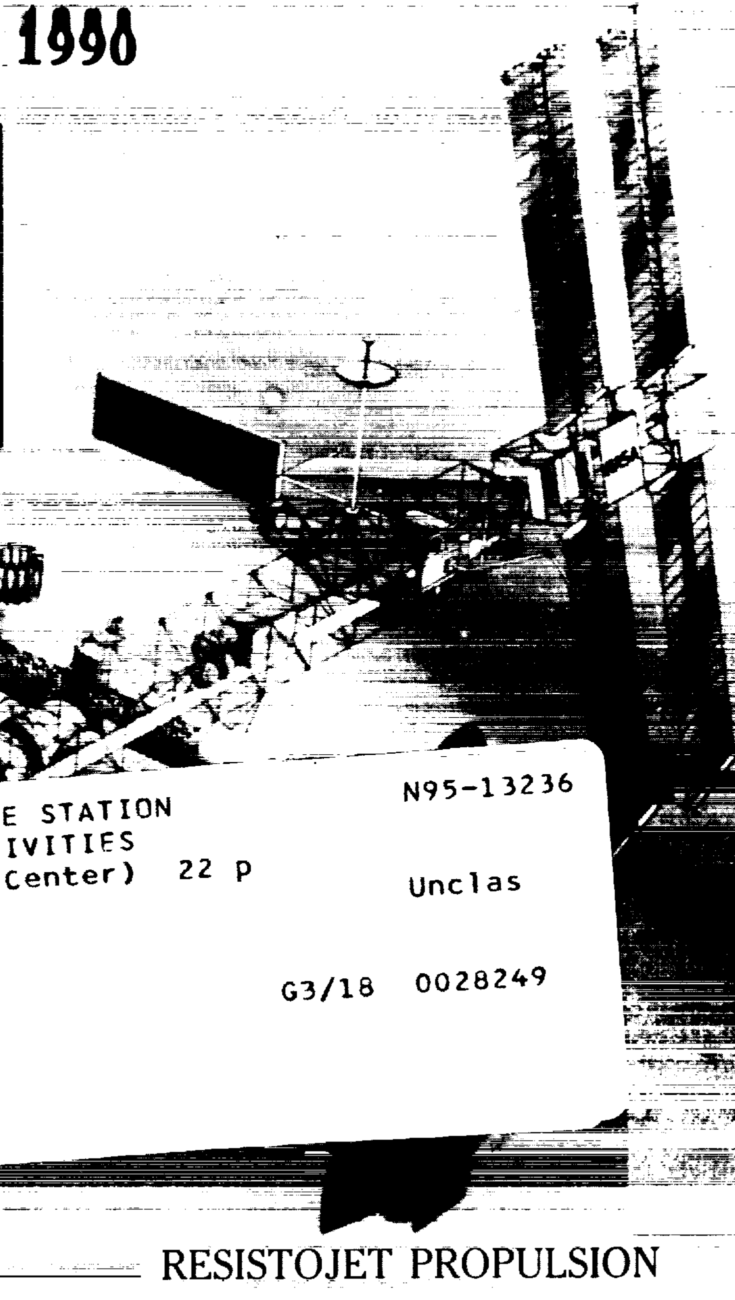
SPACE STATION FREEDOM PROPULSION ACTIVITIES

APRIL 1990



HYDROGEN-OXYGEN PROPULSION



(NASA-TM-108604) SPACE STATION FREEDOM PROPULSION ACTIVITIES (NASA. Lewis Research Center) 22 p

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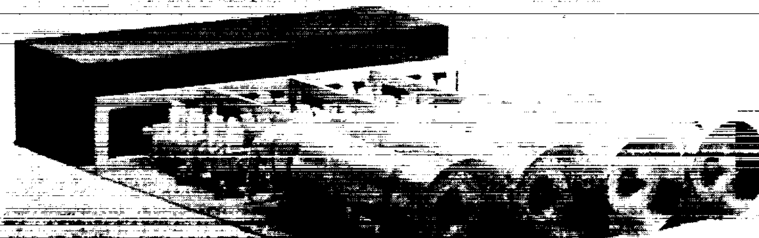
RESISTOJET PROPULSION



NASA Lewis Research Center

Foreword

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Space Station Freedom Propulsion Activities is a periodic publication that highlights technical progress and accomplishments of the NASA Lewis Research Center (LeRC) that support development of the SSF propulsion system. The objectives of these efforts are to:

- o Develop and characterize resistojet–thruster components and assemblies
- o Develop and characterize hydrogen–oxygen thruster components
- o Conduct system trade studies

Research projects primarily characterize propulsion performance and life. Other tests include environmental impacts, such as exhaust gas profiles and electromagnetic interference. Technical activities highlighted are being conducted at LeRC within the Aerospace Technology and Space Station Freedom directorates. These activities include:

- – Derivation of design analysis models
- – Trade studies of design options
- – Propulsion system impact studies
- – Component testing for characterization and design verification

This publication is intended for the information and use of organizations and personnel which share concern about Space Station Freedom propulsion. These include:

- o NASA Headquarters
- o The NASA field centers
- o Industry
- o The international community
- o Space Station Freedom working groups, particularly,
  - – Propulsion System
  - – Fluid Management Systems (FMS)
  - – Environmental Control and Life Support Systems (ECLSS)
  - – Natural and Induced External Environment

This publication includes a bibliography of LeRC reports that document the development of propulsion systems for the Space Station Freedom. The work completed on Advanced Development Phase activities is summarized in the report Space Station Propulsion Technology (October 1987, NASA TM 100108). LeRC has also been involved in two system trade studies, documented as follows: *Space Station Propulsion Analysis Study*, (June 1984, AIAA 84–1326), and *Space Station Propulsion Requirements Study*, by Boeing Aerospace (August 1985, CR 174934).

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Resistojet Life Tests

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In the case of resistojet thrusters, mechanical failure may be due to thermal fatigue which is associated with

1. operational cycling,
2. creep,
3. grain growth/boundary weakening, or
4. a combination of these.

Mechanisms two and three are associated with operating time at elevated temperatures.

Resistojet components identified for potential mechanical failure are the heater (the most likely candidate) and heat exchanger/pressure vessel. Besides mechanical failure as described above, failure in the heater element could occur due to local melting or material property changes (e.g., electrical insulation breakdown). Finally, nozzle erosion may occur and would first affect thruster performance; later it could lead to mechanical failure.

Test Objectives and Conditions

The goal of these life tests is to expand the scope and accuracy of the database providing characteristics of multipropellant resistojet thrusters, and their prime components. Even though complete life requirements and other design criteria have not been established for Space Station Freedom resistojet thrusters, life test data will be useful in the preliminary design process.

The test conditions for thruster life testing were derived from the contamination requirement for a 14–day quiescent period and the assumption that the resistojet system must exhaust all waste fluids generated onboard Space Station Freedom. It was assumed that four of the resistojet thrusters would operate simultaneously at 500 W each with a heater temperature of 1,200° C. This results in approximately three days (or less) of "on" time.

The thruster was tested with nitrogen because nitrogen closely approximates the temperature profile through the thruster of the mixed gases from Space Station Freedom, and it is readily available at the facility. Carbon dioxide was also used for some tests, because it duplicates SSF pressure conditions. The thruster operated at a nominal heater temperature of 1,200° C for three days, followed by cooling (approximately 6 hours) to ambient temperature. After the thruster reached ambient, a new cycle was begun. Every 1,500 hours the thruster was rotated 180° to minimize gravitational effects.

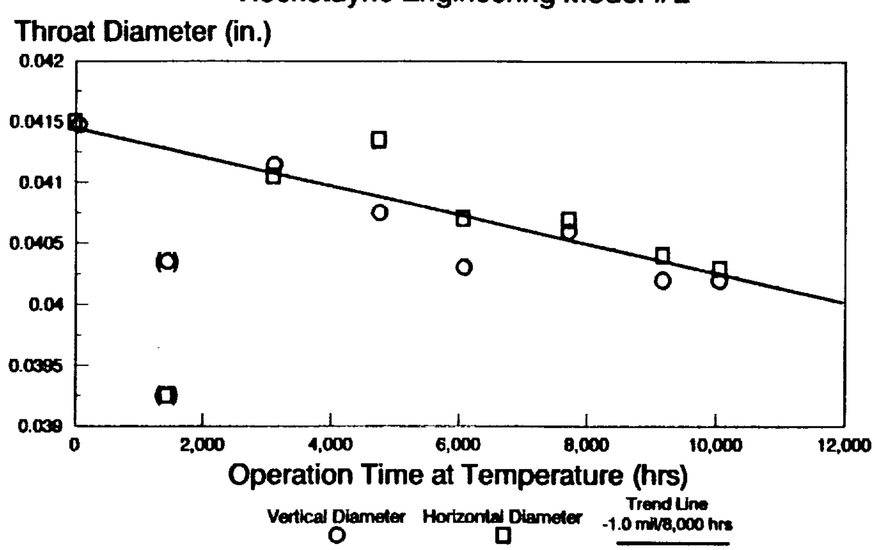
Test Results

The resistojet thruster successfully met its design goals by completing 10,036 hours and 141 thermal cycles (from room temperature to 1,000° C) of operation with no noticeable degradation and with minimal maintenance. There are no plans to dissect the thruster (to examine grain structure, surface condition, etc.), and no further tests are planned at this time. Other results are as follows:

Throat erosion was negligible. Throat diameter measurements were taken at intervals of approximately 1,500 hr using an optical comparator. The results are shown in Figure 1. No erosion was detected. Instead, there was a small but measurable reduction in diameter (approx. 1 mill in 8,000 hr), probably caused by a build–up of contaminates on the throat walls.

Figure 1: Thruster Throat Diameter Change

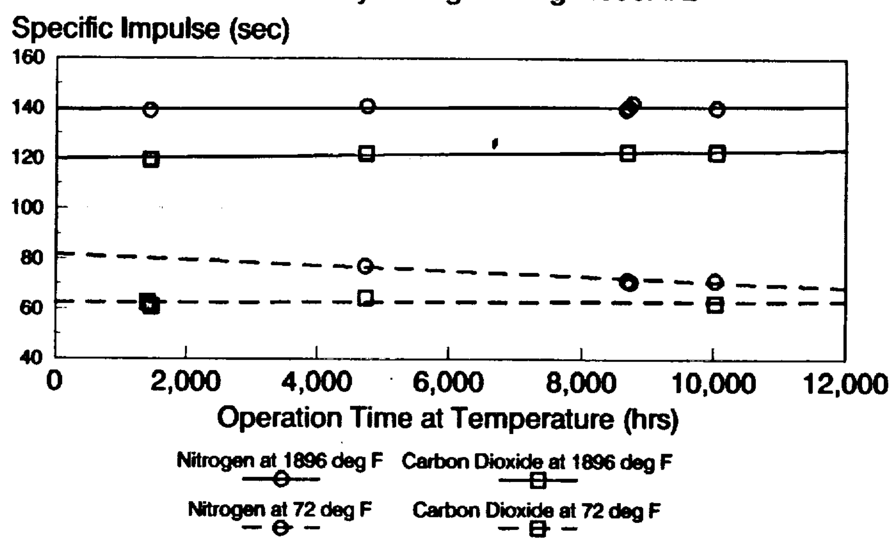
Rocketdyne Engineering Model #2



Specific impulse remained constant. The thruster was periodically removed from the life test chamber and placed on a calibrated test stand to measure thrust and mass flow rate. Specific impulse was then calculated from these data and the results are presented in Figure 2. Except for operation with room–temperature nitrogen, ISP stayed constant or increased slightly during the 10,000 hr of operation.

Figure 2: Specific Impulse Variation

Rocketdyne Engineering Model #2



AC power was required. The thruster was first operated on DC power, but it became impossible to maintain the required 500 W input to the heater. Diagnostic testing determined that insulation in the heater had become ineffective. The exact mechanism of this breakdown is not yet known. By changing to an AC power supply the insulation's effectiveness was restored, and maintaining 500 W of power to the heater was no longer a problem.

Concluding Remarks

This thruster design, by surviving 10,000 hr of life testing with thermal cycling to temperatures of 1,000°C, demonstrated both its ruggedness and reliability under conditions representing Space Station Freedom service. Further research is recommended on the following effects:

1. Effect of higher temperatures on thruster performance and durability.

2. Effect of operation with contaminated gases (like SSF waste gases).

3. Effect of long–term operation with DC power on effectiveness of electrical insulation in the heater circuit.

Resistojet Heater Calibration Test

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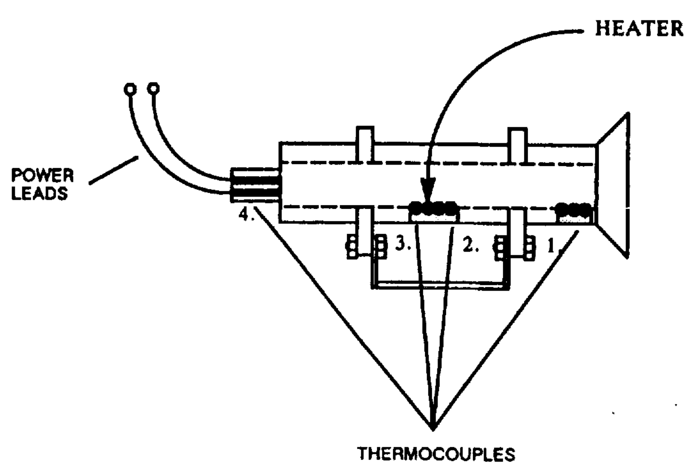
A calibration test was performed on Rocketdyne Thruster No. 1, to establish heater temperature versus heater resistance, in support of the 10,000–hr endurance test conducted at LeRC. Significant improvements in calibration test conditions included operating at a good vacuum level of 5 x 10^{-5} torr, compared to 5 x 10^{-2} torr during previous calibrations. The latter pressure level is high enough to support convective heat transfer which could adversely affect the results. The heater was instrumented with four thermocouples as shown in Figure 1. The test was run on direct current, over a temperature range from ambient to 1,100°C (2,012°F).

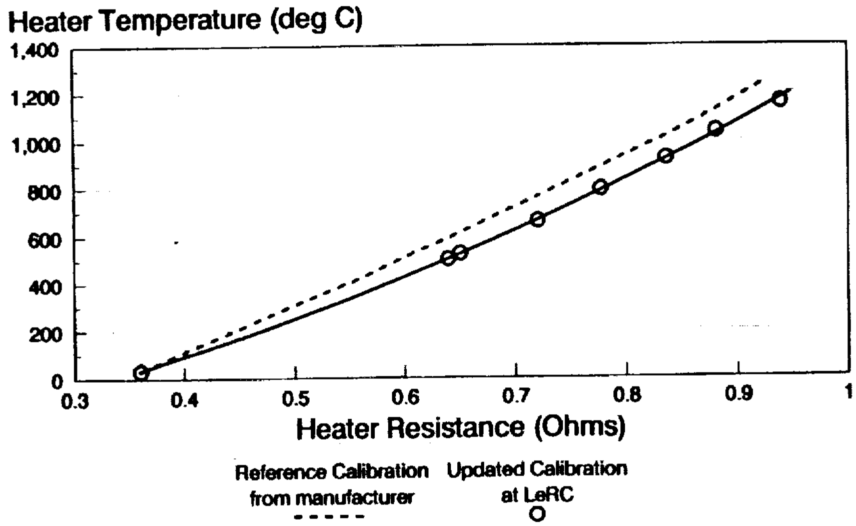
Figure 1: Diagram of Resistojet Heater
Showing Thermocouple Locations



Test Results

Figure 2 presents the results of the calibration test, compared to the reference calibration curve supplied by the manufacturer. At higher temperature levels, it is necessary to operate at somewhat higher resistances than those given by the manufacturer, in order to achieve a given heater temperature.

Figure 2: Heater Calibration Curve
for Rocketdyne Resistojet Thruster #1



Future Work

A multiple–heater life test was planned to be run in the same tank utilized for the calibration test. New turbomolecular pumps were installed, and vacuum levels on the order of 10^{-6} torr were achieved. Cold–wall temperatures down to –30° C (–20° F), were demonstrated. Because of changes in program priorities, this tank is currently scheduled for other research. However, the capability of conducting heater life tests will be maintained for the near future.

Space Station Freedom Hydrogen–Oxygen Propulsion

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The objective of the Space Station Freedom GH_2/GO_2 propulsion development program is to demonstrate durability and reliability of small gaseous H/O rockets. Primary technology goals are as follows:

- Thrust operating range from 12.5 to 31 lbf, with a design thrust of 25 lbf.
- Mixture ratio (MR) operating range from 3.0 to 8.5, with a design MR of 8.
- Specific impulse of at least 364 sec.
- Cumulative operating life equivalent to 22.2 hr at 25–lbf thrust (2,000,000 lbf–sec).

Contracts were awarded to Aerojet and Bell to build two GH_2/GO_2 thrusters with a design MR of 4. The second Aerojet thruster was re–designed for MR = 8, when the water–electrolysis propulsion system was baselined for SSF. A contract was also awarded to Rocketdyne to build two thrusters designed for MR = 8. Characterization testing of all these thrusters is continuing, in conjunction with other low–thrust propulsion research at LeRC. Life testing was put on hold when hydrazine was baselined for SSF.

Tests of Rocketdyne Designs

Testing has been performed on a series of Rocketdyne designs with coaxial injectors, in an effort to find a balance between good performance and sufficient cooling. Test results to date are summarized in Table 1.

Table 1: Hydrogen-Oxygen Thruster Test Data
Rocketdyne Designs for 25-lbf Thrust at MR = 8

Design No. (Test Site)	Oxidizer Injector Post Angle	Fuel Film Cooling	Temp. Margin at Ext. Wall (a)	Specific Impulse (b)
Prototype (MSFC)	Straight	40 %	+ 280 °F	360 sec
2 (MSFC/LeRC)	Straight	40 %	- 50 °F (est)	354 sec
3 (MSFC/LeRC)	Canted	15 %	+ 200 °F	345 sec
4 (LeRC)	Canted	7.5 %	- 100 °F (est)	348 sec
5 (LeRC)	Canted	15 %	+ 200 °F	355 sec (preliminary)

(a) Redline Temperature = 1,100 °F (b) Goal = 364 sec

The prototype thruster, with straight injector posts and 40% fuel film cooling (FFC), operated with good cooling margin and essentially met the specific impulse goal. This design contained wires in the cooling channels for improved heat transfer. In Design No. 2, the wires were removed in an effort to improve performance. Injector post angle and FFC percent were the same as the prototype. However, wall temperatures hit the redline before reaching steady state at MR = 8. Design No. 3, with its canted injector posts and 15% FFC, provided sufficient cooling but had lower performance than the first two designs. Reducing the FFC to 7.5% (Design No. 4) caused over–heating.

Design No. 5 (the latest design generation) is another with a canted injector and 15% FFC, but it has an oxygen annulus similar to the prototype and is fabricated to tighter tolerances. Preliminary data indicate that this latest injector has somewhat higher performance than Design No. 3, while maintaining chamber temperatures below the redline.

An integral exciter/igniter, used in the space shuttle main engine and baselined for Rocketdyne's flight–type thruster, was successfully demonstrated during these thruster tests. A J–2 exciter was used previously.

Future Research

The two Rocketdyne 15% FFC, canted injectors (Designs 3 and 4), will be shipped to JSC for testing in a water–electrolysis–system testbed. Testing of a Rocketdyne resonance igniter (developed with internal funding) is planned for June 1990 at LeRC. The resonance igniter is designed to dynamically heat hydrogen above the auto–ignition temperature of oxygen gas, eliminating the need for an electrical ignition system.

The second Aerojet thruster is serving as a testbed article for low–thrust computational fluid dynamics (CFD) research. Characterization testing was performed during the summer of 1989 and the winter of 1989/90, and more testing is planned for the summer of 1990.

In the summer and fall of 1990, 5–lbf and 25–lbf radiation–cooled thrusters will be tested at LeRC. These thrusters are fabricated from high–temperature, oxidation–resistant materials that allow operation at internal wall temperatures up to 4,000 °F. The thermal margins provided by these advanced thrusters offer significantly improved performance and/or longer life, which could benefit the evolution of space station propulsion systems.

Fundamental Studies of Low Reynolds Number Nozzle and Plume Flows

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and

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Work is progressing in this cooperative program between the Lewis and Ames Research Centers, which has the following objectives:

- o To determine the validity of analyses of rarefied flow and identify possible deficiencies in applying continuum methods to such flows.
- o To verify with experimental data the prediction of plume flowfields from small thrusters in space.
- o To identify possible plume effects on satellite surfaces, such as contamination, disturbance torques, and heat loading.

Thruster Configuration

The initial case for both analysis and experiment is a conical nozzle with a throat diameter of 0.318 cm and an area ratio of 100. The working gas is CO_2 at a stagnation pressure of 7,700 Pa (N/m^2) and a temperature of 1,000 K, which gives a Reynold's Number (based on throat diameter) of about 1,000. The flow for this case is quite viscous and hence requires analysis by the Navier–Stokes equations in the continuum regime. The flow, however, attains considerable rarefaction at the nozzle exit where the continuum assumption becomes questionable.

Two Analytical Methods

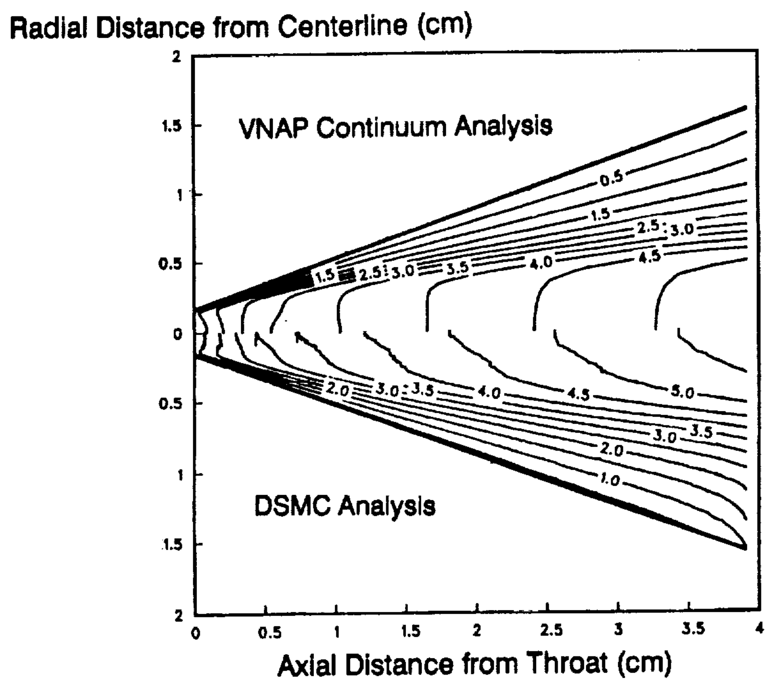
The approach of this study is to analyze the flow with both a Navier–Stokes code and a code based on a direct simulation Monte Carlo (DSMC) scheme. With the DSMC method, all physical phenomena are modelled on the molecular level, including (as the Monte Carlo name implies) a statistical analysis of molecular collisions and trajectories.

The continuum analysis has been conducted at LeRC with the VNAP (Viscous Nozzle Analysis Program). Plans are to use an alternate program for a more accurate Navier–Stokes solution. The DSMC analysis is being performed at ARC, using the Ames/Stanford Particle Kinetic (ASPK) code. Results from the continuum analyses are used as starting or boundary conditions for the DSMC analyses.

Analytical Results

Results from each of the numerical schemes are illustrated in Figure 1, which shows contours of constant Mach Number. Although the DSMC computation needs to be refined, the results do compare reasonably well (at least qualitatively) with those from the continuum analysis [1]. Deviations do occur at the nozzle lip (as expected), because of the degree of rarefaction of the gas at that point in the flow, and because of the somewhat artificial boundary condition that is applied to the flow in the continuum analysis in the subsonic region at the exit near the nozzle wall.

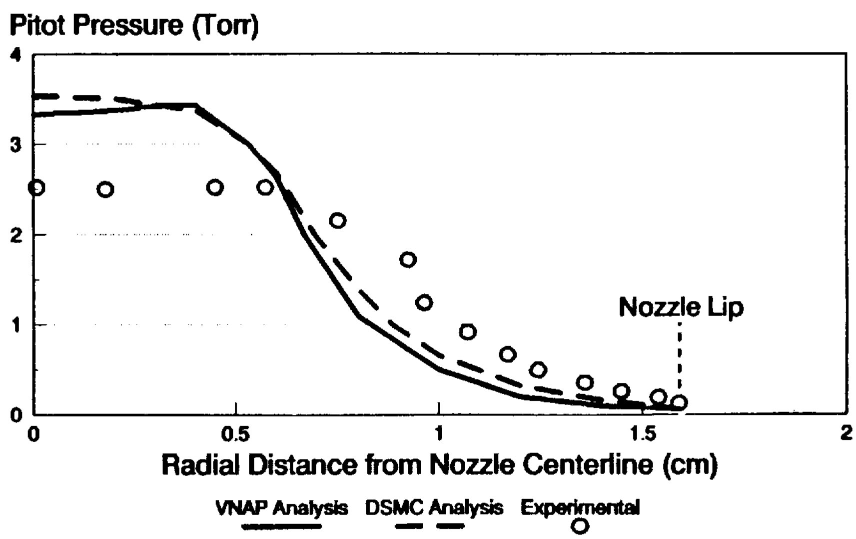
Figure 1: Calculated Velocities Inside Nozzle
Lines of Constant Mach Number



Test Verification

To complement the theoretical analyses, experimental data have been taken recently in the large vacuum tank (Tank 5) of the Electric Propulsion Laboratory at LeRC. These data consist of pitot pressure measurements taken near the exit plane of the nozzle. A typical pitot scan is shown by the data points in Figure 2. As an initial comparison, calculated pitot pressure distributions are shown for the exit plane (see Fig. 1), according to the two analytical models discussed earlier. Comparison between computed and experimental results are within reason, although there were slightly different conditions of axial location, total pressure, and temperature between experiment and analysis. The computer codes will be rerun to match experimental conditions.

Figure 2: Experimental and Analytical Pressures
Distributions Across the Nozzle Exit Plane



A thorough pitot–pressure survey will be taken to identify flow direction in the nearfield plume where pressure measurements are valid. Measurements of mass deposition with a quartz microbalance will also be taken in the plume farfield to determine the nature of plume expansion. These measurements will be used to validate computed results of the plume from the molecular model.

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Basic Research in Arc–Jet Technology

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Arc–jets, like resistojets, convert electrical energy into kinetic energy to produce thrust. In an arc–jet, however, electric current is run directly through the ionized propellant between a cathode and an anode (nozzle) to increase its enthalpy, unlike a resistojet which employs a heat exchange mechanism such as a heating element. Direct "heat injection" results in a higher specific impulse, I_{sp}, in arc–jets compared to resistojets. This additional I_{sp} comes with some penalties, however. A considerable amount of electrical energy is spent to ionize the propellant and stays trapped in the "frozen flow mode". Also, the initial ionization requires a power conditioning unit with high–speed current regulation and high–voltage ignition capabilities, both of which translate into extra mass.

Background

Arc–jet research dates back to late 1950's and early 60's. Most of this work ceased during the 70's because of on–board power limitations and electrode erosion problems. With the advent of solar arrays which can produce more than 30 kW and the prospect of on–board nuclear power, arc–jets came back onto the low–thrust propulsion scene in the 80's. Using strong swirl–stabilization, electrode erosion rates were lowered to a reasonable level. Smaller and more effective power conditioning units have been built. With these advances the goal of long–term operation appears feasible, and arc–jets have become strong candidates for satellite station–keeping missions.

Now that certain basic problems have been solved, we face the following questions as we are developing the next generation of arc–jets:

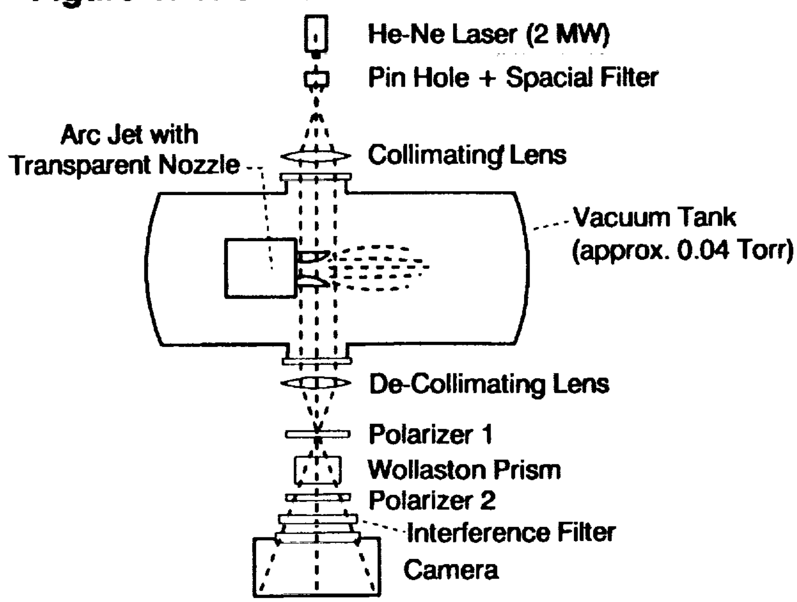
- Within an arc–jet, how does the plasma behave in the outer and colder flow? How thick is the arc diameter? How does it attach to the anode? How hot does it get?
- How does clogging of the nozzle throat (by the arc) affect overall performance?
- What are the nozzle geometry and flow–field effects on electrical field strength and also on heat transfer rates?
- Is there an optimum swirl rate?

Optical Diagnostics

In order to answer the first two types of questions, one is compelled to resort to optical diagnostic techniques which can range from the old Schlieren method to the more–recent Wollaston Diffractiometer (WD) and spectroscopic measurements. The WD technique [1] is especially useful for visualizing isodensity lines in flow fields with the steep thermal gradients characteristic of arc–jets.

Recently, a rectangular, transparent arc–jet and a Wollaston Diffractiometer have been built at LeRC for optical diagnostic studies (Fig. 1). This facility enables one to observe the inside of the nozzle in addition to the nozzle plume. The arc–jet currently under test employs a multi–expansion nozzle to create high values for the local mass flux, and three different swirl injectors, to study the effect of vortex strength on arc–jet performance.

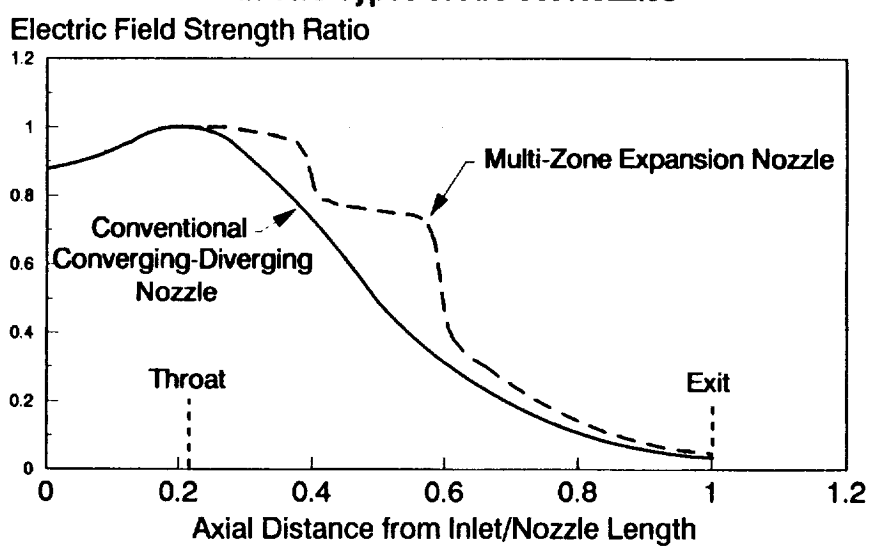
Figure 1: The Wollaston Diffractiometer Facility



Multi–Zone Expansion Nozzles

Recent studies [2] show that the nozzle geometry is one important determinant factor in arc–jet performance. It has been formulated that the electric field strength, E, is a strong function of the nozzle mass flux, ρu. This functional relationship can be expressed in a simplified way as $E = k\ (\rho u)$. Here k is not a constant but a function of plasma temperature and acceleration. Thus, new nozzle designs which yield higher local mass fluxes should increase electric field strength and, therefore, the Ohmic heating of the propellant. One such design is a multi–zone expansion nozzle which theoretically would result in higher electric field strength than a conventional converging–diverging nozzle, as depicted in Figure 2.

Figure 2: Electric Field Strength Distribution
In Two Types of Arc Jet Nozzles



Swirl Stabilization

Vortices induced by the swirl action diminish the axial pressure gradient inside the nozzle, which is the main driving force for the propellant. There may be an optimum swirl rate which reduces electrode erosion and at the same time results in a reasonable axial pressure gradient. By employing different injectors of various vortex strengths in the LeRC arc–jet facility, such an optimum swirl rate could be determined experimentally.

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3. Whalen, M.V.; Grisnik, S.P.; and Sovey, J.S.; *Compatibility Experiments of Facilities, Materials and Propellants for Electrothermal Thrusters*, NASA TM-86956, April 1985

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